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RESEARCH MEMORANDUM

PRESSURE RECOVERY AT SUPERSONIC SPEEDS THROUGH ANNULAR DUCT INLETS SITUATED IN A REGION OF APPRECIABLE BOUNDARY LAYER. II - EFFECT OF AN OBLIQUE SHOCK WAVE IMMEDIATELY AHEAD OF THE INLET

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CLASSIFICATION CANCELLED

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

PRESSURE RECOVERY AT SUPERSONIC SPEEDS THROUGH ANNULAR DUCT INLETS SITUATED IN A REGION OF APPRECIABLE BOUNDARY LAYER. II — EFFECT OF AN OBLIQUE SHOCK WAVE IMMEDIATELY AHEAD OF THE INLET

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SUMMARY

A model having a ramp that produced an oblique shock wave in front of an annular duct inlet was tested at Mach numbers between 1.36 and 2.01 to determine the effect of reducing the entrance Mach number on the total-pressure recovery after diffusion. The results of the tests showed that the maximum total-pressure recovery of the model investigated was considerably improved through the addition of a ramp. However, separation of the boundary layer limited the extent of the reduction of the entrance Mach number.

For the configurations tested, a 15° annular ramp was the best; the highest maximum total-pressure recovery attained was four-fifths of that of a normal shock wave occurring at the free-stream Mach number.

INTRODUCTION

The design of a fuselage for a supersonic aircraft may necessitate the use of a long, slender body of revolution with a ram-jet or a turbo-jet engine located in the stern. In order to supply air to the engine, a duct having an annular entrance situated some distance behind the nose of the fuselage may be desirable because it may have a simpler arrangement of internal members and thereby reduces the pressure losses caused by the duct system. An inlet in such a location will receive a considerable amount of boundary-layer air, and according to a previous experimental investigation the total-pressure recovery may be relatively low. (See reference 1.) This

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poor pressure recovery is attributable to the thickening and eventual separation of the boundary layer ahead of the inlet caused by the interaction between the boundary layer and the compression in the induction system.

In the case of a diffusor with a supersonic inlet, a portion of the compression occurs through a normal shock wave near the entrance. Since the tendency of the boundary layer to thicken and to separate increases with the severity of this compression, the Mach number ahead of the inlet should be as near unity as possible in order to minimize the losses through the shock wave and to reduce the intensity of the adverse pressure gradient imposed upon the boundary layer.

This report presents the results of an experimental investigation of the effect on total-pressure recovery after diffusion of placing a ramp ahead of an annular inlet to reduce the entrance Mach number by means of an oblique shock wave.

SYMBOLS

H total pressure

M Mach number

A area

m rate of mass flow

The following subscripts indicate the station of the measured quantity:

o free stream

1 duct entrance

3 settling chamber

4 exit throat

APPARATUS AND METHODS

The tests were performed in the Ames 8- by 8-inch supersonic wind tunnel in the range of Mach numbers between 1.36 and 2.01; the

Reynolds numbers, based on the length of the model forebody, were between 2.23 and 3.04 million, respectively. The models were tested at zero angle of attack. A description of the tunnel, the model support, and the auxiliary equipment may be found in reference 1.

With the exception of the entrance ramp, the model tested was similar to model B of reference 1. A photograph of the model is shown in figure 1, and the dimensions are given in the drawing of figure 2. The forebody consisted of a 10-caliber ogival nose followed by a cylindrical section. The inlet was an annular opening, the area of which was 28 percent of the total cross-section area at the inlet station. Immediately ahead of the duct entrance, the cylindrical forebody surface was flared to form a ramp which consisted of an annular wedge. The angle that this wedge made with the longitudinal axis of the model was altered by reducing the length of the ramp while the height remained the same. The angles tested were 5°, 10°, 15°, $17\text{-}1/2^\circ$, 20°, and 30°. Model B of reference 1 was used for the model having a 0° ramp angle. The inlet was the entrance to a subsonic diffusor having a 10.5° equivalent-cone angle. The diffusor exhausted into a settling chamber which had a variable-area outlet that was adjustable from outside of the tunnel.

RESULTS AND DISCUSSION

As shown in figure 3, a considerable improvement in the total-pressure recovery is attained after diffusion with an annular duct inlet as a result of placing a ramp ahead of the inlet to create an oblique shock wave. The greatest recovery is attained by the model having a 15° ramp. At a Mach number of 1.36, the increase in the maximum total-pressure ratio $(H_s/H_O)_{max}$ obtainable with this model is 17-1/2 percent of the recovery of model B of reference 1; at a Mach number of 2.01, this improvement is 63-1/2 percent. The total-pressure recovery with the 15° ramp model is about four-fifths of that of a normal shock wave occurring at the test Mach number.

The variation of maximum total-pressure ratio with the ramp angle at several Mach numbers is shown in figure 4. As in the case of the slotted twin-scoop inlets of reference 4, there is a large gain in the pressure recovery as the ramp angle is increased to about 10°. The rate of gain decreases as the angle approaches 15°. At larger angles, the flow becomes unsteady and the recovery decreases. The ramp angle, and hence the strength of the oblique shock wave, must be limited to a relatively low value because the compression caused by the stream deflection results in thickening of the boundary layer. The static pressure rise occurring in the duct causes additional thickening of the boundary layer, and at maximum recovery the combined compression is great enough to force the boundary layer to separate.

The model having a 15° annular ramp produces about the same maximum total-pressure ratio as a twin-scoop inlet that encloses 61.5 percent of the forebody circumference and has a 5° ramp. (See reference 2.) Since the maximum pressure recovery occurs just before the boundary layer separates, the pressures adverse to the flow of the boundary-layer air are equal for these two inlets. The twin scoops receive a larger amount of high energy air than the annular inlet and therefore produce a stronger normal shock wave that develops the allowable adverse pressure with a smaller deflection of the stream, that is, at a lower ramp angle. When the height-to-width ratio of the twin-scoop inlet is increased from 0.3 to 0.75 and the inlet encloses 37.2 percent of the forebody circumference, the maximum recovery improves 1 to 4 percent and occurs at a ramp angle of 12° (reference 3). As the width of the ramp is decreased, the transverse pressure difference becomes more effective in thinning the boundary layer by diverting it from the inlets. The maximum recovery increases because of a delay in separation possibly caused by a thinner boundary layer on the narrow ramp.

The schlieren photographs of figure 5 show the flow about the model having various ramp angles. These photographs were taken at relatively large values of the outlet-inlet-area ratio in order that the pressure in the diffusor would not have an appreciable effect on the boundary layer flowing into the inlet. The flow is of the same nature as that described in reference 4 where the effects of increasing ramp angle on the pressure recovery through a slotted twin-scoop inlet are described. When the ramp angle is smaller than about 12°, an oblique shock wave originates from the ramp leading edge. At greater angles, the compression resulting from the deflection of the stream is sufficiently large to cause the thickening of the boundary layer ahead of the ramp. As a result, the boundary layer bridges the break in the surface, thereby deflecting the stream through a smaller angle and causing less compression than would occur if the flow followed the surface. Measurements of the position at which the oblique wave occurs for the models having a 15°, $17\text{-}1/2^{\circ}$, 20°, or 30° annular ramp indicate that the intensity of the wave changes only slightly and is roughly equivalent to that which would be produced by a 12° ramp. At ramp angles greater than 15°, the flow through the air-induction system becomes unsteady and the total pressure in the settling chamber fluctuates considerably. This oscillation was observed to be irregular and possibly was caused by eddying flow over the ramp; as a consequence, the average total-pressure recovery decreases, and the experimental data show considerable scatter.

A typical variation of the total-pressure recovery with the mass-flow ratio[1] is shown in figure 6 for a model with a 15° ramp. It is seen that the recovery of total pressure is very sensitive to a change in the mass flow near the maximum total-pressure ratio. The introduction of an oblique shock wave apparently delays the separation of the boundary layer. Test observations reveal that when the separation finally does occur, it appears suddenly and causes an abrupt change in recovery. This phenomenon is indicated by the sharply cusped curves of figure 6.

CONCLUSIONS

Tests at Mach numbers between 1.36 and 2.01 of an annular duct inlet having a ramp to create a single oblique shock wave to reduce the entrance Mach number have shown the following effects:

1. The pressure recovery attainable with an annular duct inlet situated in a region of appreciable boundary layer is considerably improved through the reduction of entrance Mach number by an oblique shock wave occurring upstream of the inlet. The allowable intensity of this shock wave is limited to a relatively small value because of its tendency to thicken the boundary layer.

2. The oblique shock wave produced by a 15° annular ramp is optimum for the configuration tested. Use of ramp angles greater than 15° resulted in unstable flow and thus reduced the pressure recovery.

3. The maximum total-pressure recovery attained is approximately four-fifths of that of a normal shock wave occurring at the free-stream Mach number.

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[1]The mass-flow ratio is defined as the mass of air that enters the duct divided by the mass that flows through a tube of the same area as the inlet in the free stream.

REFERENCES

1. Davis, Wallace F., Brajnikoff, George B., Goldstein, David L., and Spiegel, Joseph M.: An Experimental Investigation at Supersonic Speeds of Annular Duct Inlets Situated in a Region of Appreciable Boundary Layer. NACA RM No. A7G15, 1947.

2. Davis, Wallace F., and Goldstein, David L.: Experimental Investigation at Supersonic Speeds of Twin-Scoop Duct Inlets of Equal Area. I – An Inlet Enclosing 61.5 Percent of the Maximum Circumference of the Forebody. NACA RM No. A7J27, 1948.

3. Davis, Wallace F., and Edwards, Sherman S.: Experimental Investigation at Supersonic Speeds of Twin-Scoop Duct Inlets of Equal Area. III – Inlet Enclosing 37.2 Percent of the Maximum Circumference of the Forebody. NACA RM No. A8E04, 1948.

4. Davis, Wallace F., and Goldstein, David L.: Experimental Investigation at Supersonic Speeds of Twin-Scoop Duct Inlets of Equal Area. II – Effects of Slots Upon an Inlet Enclosing 61.5 Percent of the Maximum Circumference of the Forebody. NACA RM No. A8C11, 1948.

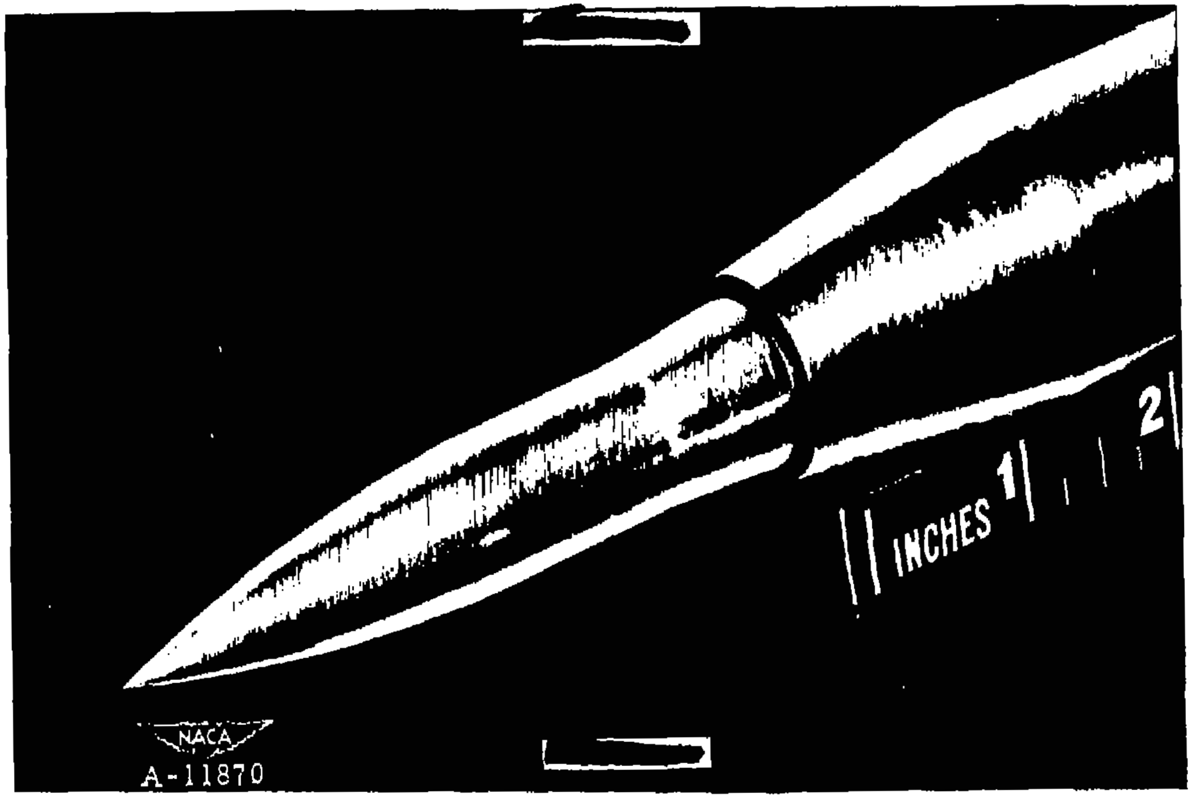

Figure 1.– Duct inlet with annular entrance and 5° ramp.

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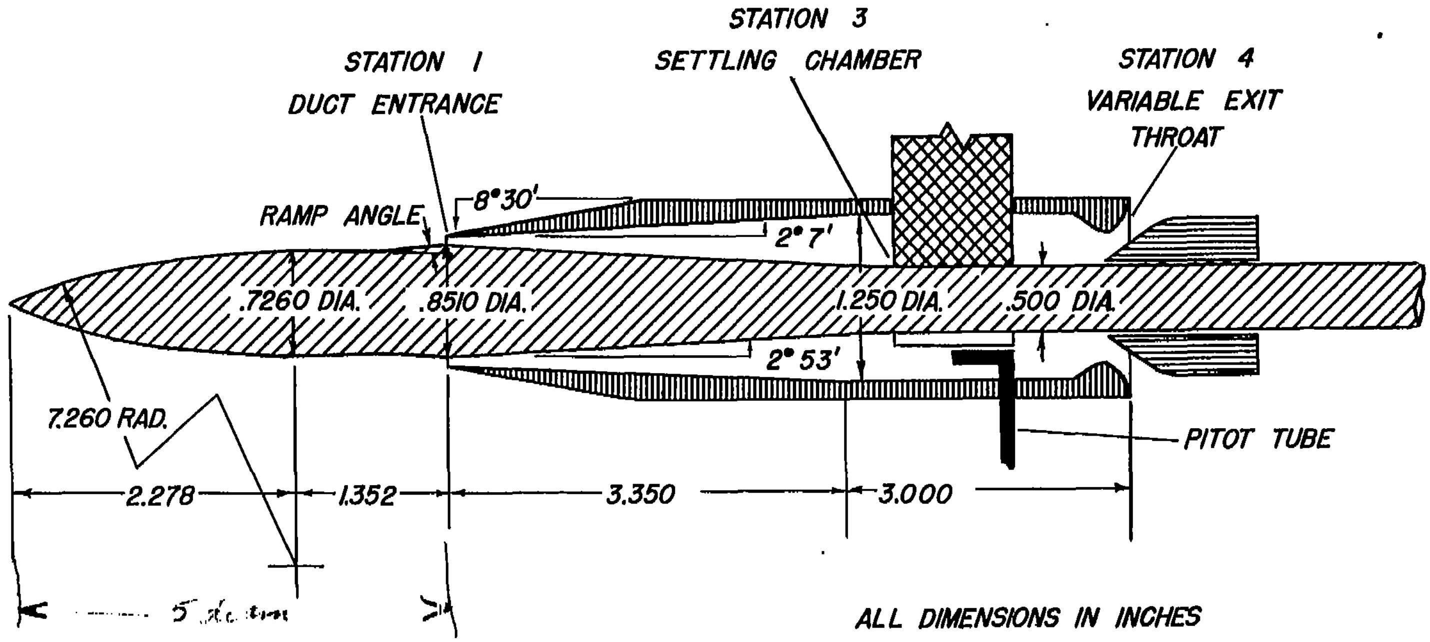

ALL DIMENSIONS IN INCHES
ENTRANCE AREA = .2209 SQ. IN.

FIGURE 2. —MODEL DIMENSIONS.

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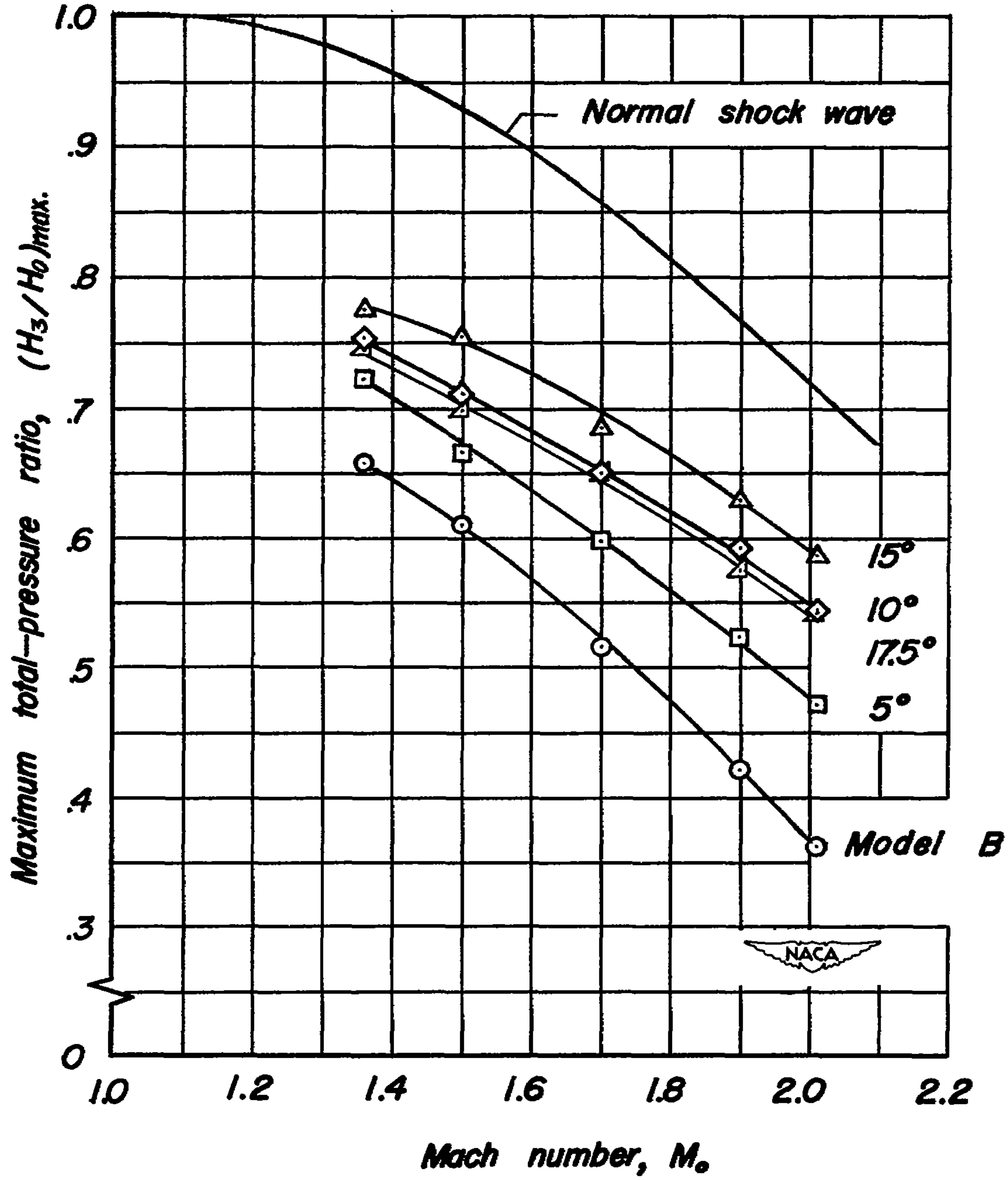


Figure 3.—Variation of maximum total-pressure ratio with free-stream Mach number and inlet ramp-angle.

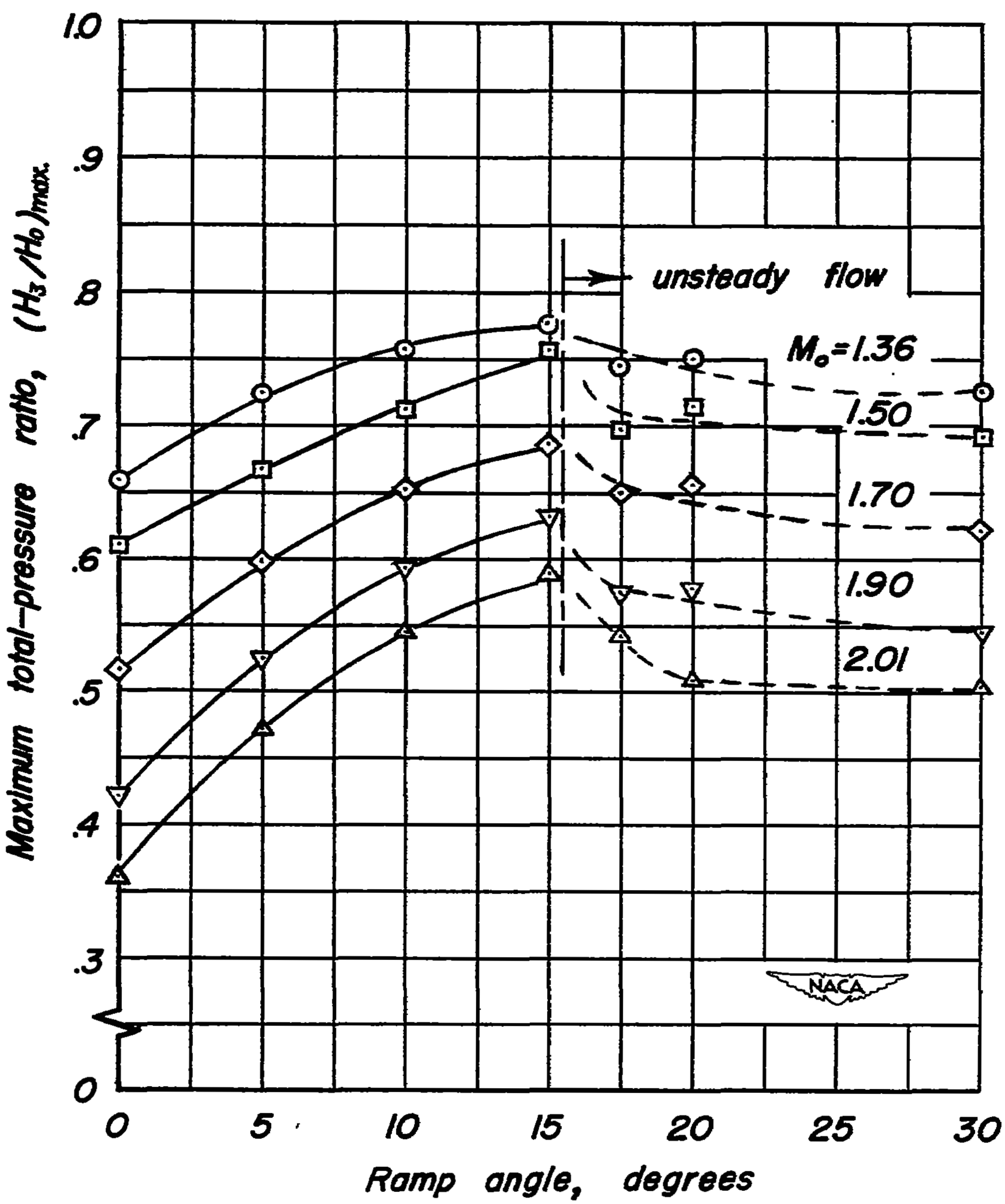
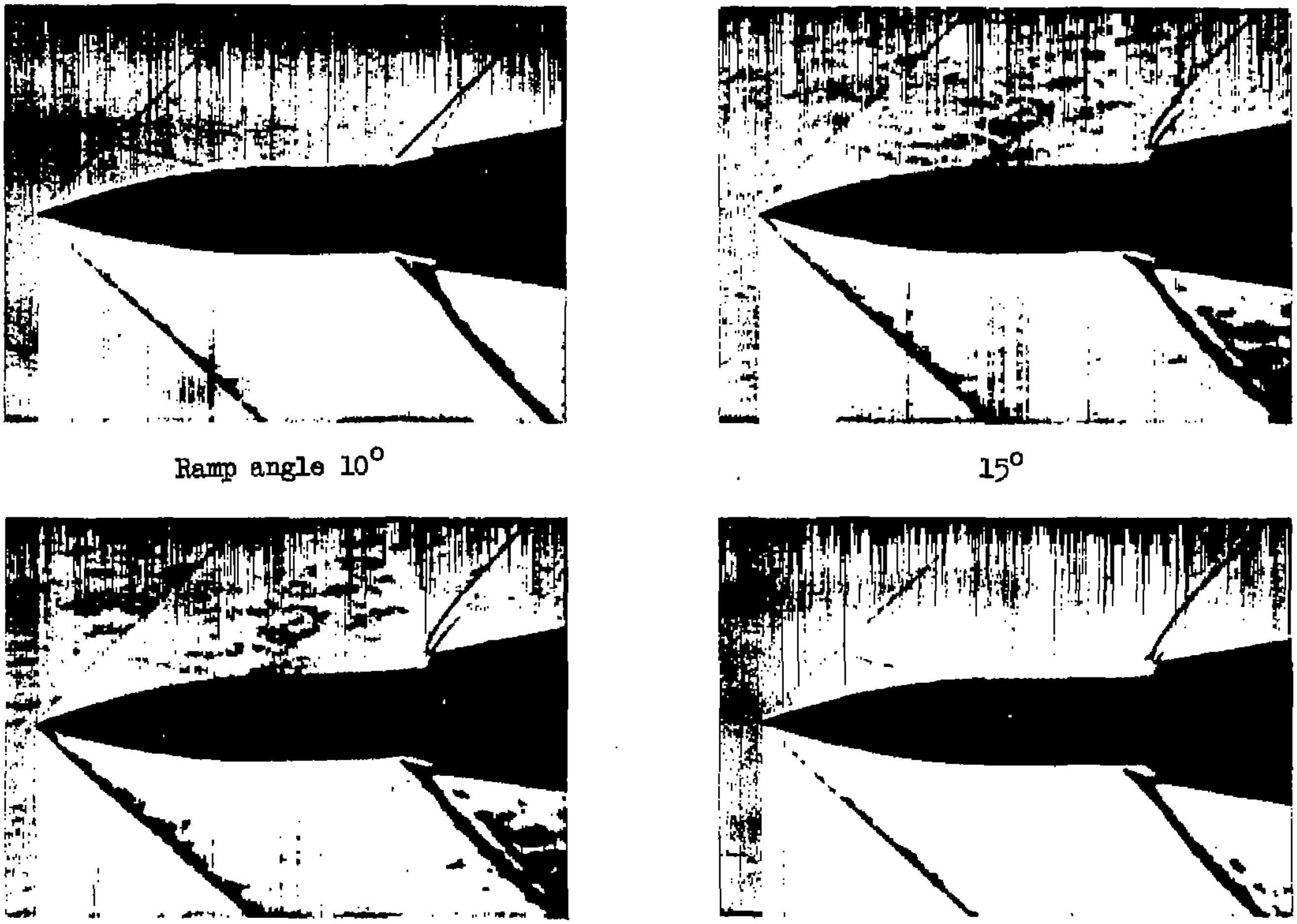


Figure 4.— Variation of maximum total-pressure ratio with ramp angle at several Mach numbers.



Ramp angle 10°

15°

20°

30°

Figure 5.– Schlieren photographs of flow about models having various ramp angles at high outlet–inlet-area ratios. Mach number = 1.70.

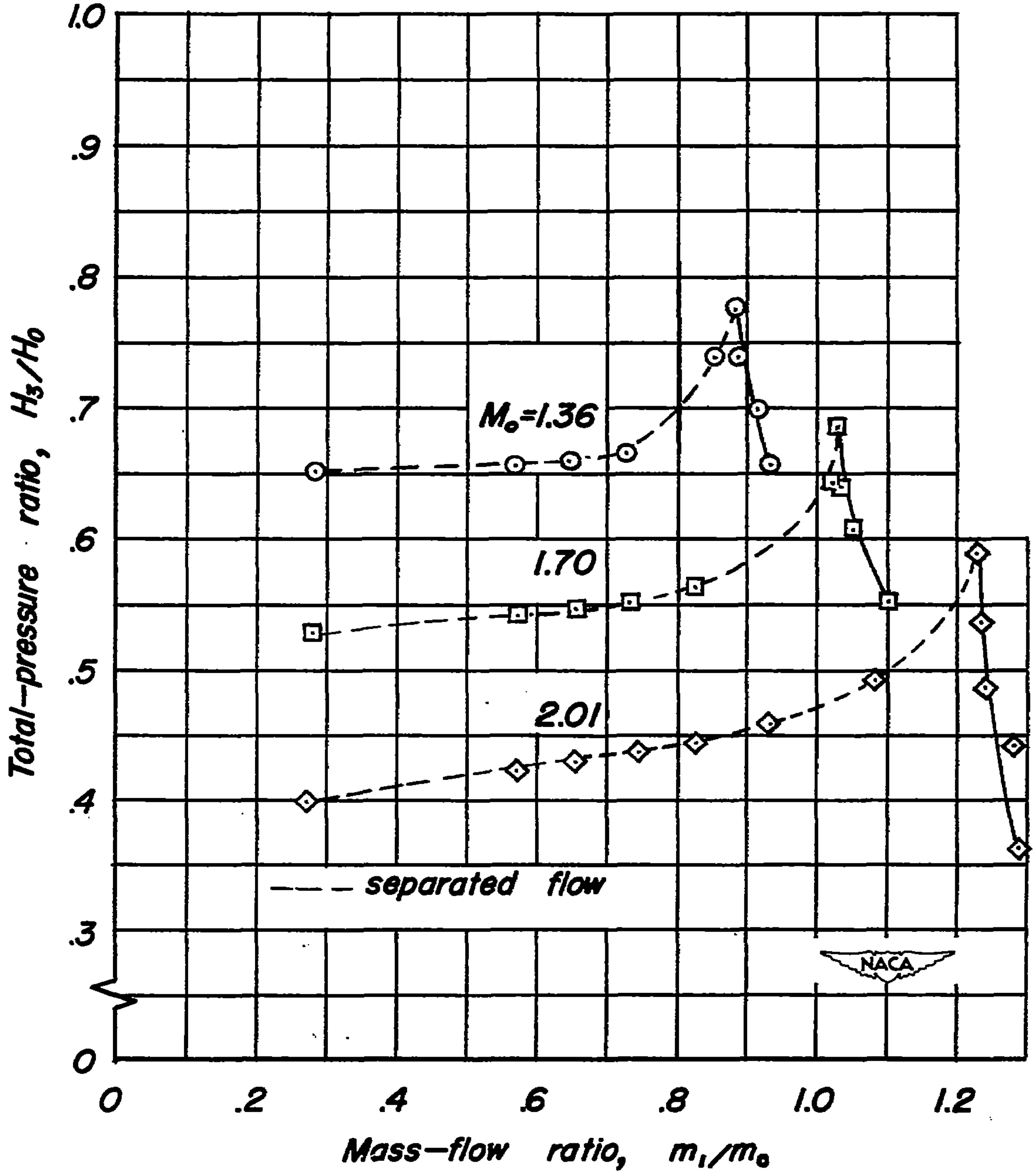


Figure 6.—Variation of total-pressure ratio with mass-flow ratio for the inlet having a 15° ramp.

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